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DESIGN METHODOLOGY AND LIFE ANALYSIS OF POSTBUCKLED METAL AND COMPOSITE PANELS: DESIGN GUIDE

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The objectives of this program were to develop an experimentally validated analysis capability and simple to use design procedures for curved metal and composite postbuckled panels loaded in compression or shear. The program plan was to first review the available analysis methods for postbuckled panels and then extend or modify these to develop a design methodology for curved postbuckled panels. This methodology was used in designing curved panels for a test program to generate design validation and fatigue life data.

The program was performed in four tasks. Task I consisted of selecting analysis methods and design procedures for postbuckled metal and composite panels. The design methodology selected was semiempirical in nature and based on classical methods for metal panels with some modifications made for application to composite materials. The analysis procedures were coded in computer programs that can be used as efficient design tools. A series of tests on curved metal and composite panels were conducted in Task II to assess the accuracy

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of the design procedures. The results from Task II showed that the semiempirical strength predictions were conservative for composite panels by approximately 30 percent. In Task III, nonempirical analysis methods based on the principle of maximum potential energy were developed to predict the displacement and strain fields in curved metal and composite panels. These predictions were compared against the test data generated in Task II. The predictions showed the same trends as the measured data. However, the numerical values could not be accurately matched. Additional work necessary to enhance the predictive capability was identified. Under Task IV, the semiempirical design methodology was documented in a preliminary Design Guide for postbuckled metal and composite panels.

The significant conclusions of the program and details pertaining to activities performed under the various tasks are documented in this final report designated as Volume I. The computer programs are documented in Volume II - Software Documentation. The preliminary Design Guide forms Volume III of the documentation for this contract.

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PREFACE

The work documented in this report was performed by Northrop Corporation, Aircraft Division, Hawthorne, California under Contract F33615-81-C-3208 sponsored by the Air Force Wright Aeronautical Laboratories, Flight Dynamics Laboratory, AFWAL/FIBE. The work was performed in the period from September 1981 through August 1985. The Air Force Program Monitors were Capt. M. L. Becker (September 1981 to June 1983), Capt. M. Sobota (June 1983 to February 1985), and Lt. P. Alsup (February 1985 onwards) who reviewed and suggested improvements to the report.

Dr. B. L. Agarwal was the Northrop Program Manager and Principal Investigator until June 1983. From June 1983 onwards Dr. R. B. Deo was the Principal Investigator. The following Northrop personnel also contributed to the performance of the contract in their respective areas of responsibility:

E. Madenci/Dr. N. J. Kudva	Analysis
F. Uldrich	Specimen Fabrication
M. Kerbow	Testing
R. Cordero	Data Analysis/Graphics
K. H. Gonzalez/B. Tuzzolino	Documentation

The results of the program were used to develop a preliminary design guide for postbuckled structures. The Design Guide (Initial Release) is published separately as Volume III.

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SECTION 1

INTRODUCTION

1.1 PURPOSE, SCOPE AND ORGANIZATION

The purpose of the Design Guide is to document a step-by-step, easy to use design methodology for postbuckled flat or curved panels loaded in compression or shear only.

This release of the Design Guide covers static design and analysis methods for flat and curved panels loaded either in compression or in shear. Stiffened panels made of composites as well as metals are addressed. The emphasis in this Guide is on illustrating the iterative design procedure and on demonstrating the use of special purpose computer programs written to accomplish the design task. Analysis details are kept to a minimum since a more complete documentation is given in Reference 1. The analytical expressions presented in the Guide are those that need to be used in addition to the programs. Procedures for executing the computer programs are documented in Reference 2. An attempt has been made to maintain commonality in the design approach for metal and composite panels. Differences in design considerations for the two material types, e.g., failure modes and the anisotropic nature of composites, are highlighted where appropriate. Panels under shear loading and under compression loading are addressed in two separate sections.

1.2 GENERAL CHARACTERISTICS OF POSTBUCKLED PANELS

Stringer or longeron and frame stiffened panels are widely used in aircraft construction. In many of these stiffened panel applications, particularly for fuselage structures, significant weight and cost savings can be realized if the skin or web between the stiffeners is permitted to buckle well below the design limit load. The weight savings advantage in such a design is

a direct result of the ability to use thin skins and space the stiffeners farther apart. The reduction in the number of stiffeners that results from a wider spacing also translates into lower manufacturing costs.

The load carrying capability of stiffened panels after skin buckling is due to the redistribution of a majority of the applied load into the discrete stiffeners and the remainder into the skin, assuming that the skin is continuously connected to the stiffeners. By appropriate design of the stiffeners, therefore, the load carrying capacity of postbuckled panels can be enhanced to several times the skin initial buckling load assuming failure occurs by stiffener crippling.

The structural response of postbuckled stiffened panels depends on the nature of loading and the panel geometry, i.e., whether the panel is flat or curved. The postbuckling behavior of compression panels is characterized by the appearance of sinusoidal buckles in the skin between stiffeners accompanied by a simultaneous increase in the fraction of load resisted by the longitudinal stiffeners (stringers). After initial buckling, the applied compression load is carried by the stringers and a small effective width of the skin adjacent to the stringers. As the compression load is increased beyond the initial buckling load, the buckles in the skin become deeper and may also change in number. If the panels are made of metal, eventual failure can occur in several possible modes such as permanent set in the skin, stringer crippling, stringer yielding or Euler buckling of the panel as a whole. For fiber-reinforced composite panels where the common design practice is to cocure the stiffeners with the skin, panel failure can occur by stiffener skin disbonding, stringer crippling or Euler buckling of the entire panel.

The characteristic response of postbuckled panels under shear loading is nearly identical to that of partial tension field beams. At initial buckling, the skin in shear panels buckles into diagonal folds. The angle of

these diagonal folds depends on the panel aspect ratio and curvature. After initial buckling, the applied shear load is resisted by axial loads induced in the stringers (chords) and the frames (uprights) as a result of the diagonal tension in the buckled skin. The angle of the folds is determined by the direction of the diagonal tension component in the skin resulting from the applied shear. The possible failure modes in metal shear panels are permanent set in the skin, forced crippling of the stringers and/or frames due to the axial compression load and the buckles in the skin, or stiffener yielding. In composite panels, failure can occur by skin rupture due to the diagonal tension stress, forced crippling of the stiffeners, or by disbonding of the skin and the stiffeners. In addition, irrespective of the type of material used, excessive stiffener flexibility may lead to shear buckling of the panel as a whole.

The complexities of load redistribution after skin buckling and existence of multiple failure modes, make the use of rigorous analysis techniques to design postbuckled structures prohibitive. The methods presented in the Design Guide, therefore, are semiempirical and intended to facilitate rapid iterative design.

SECTION 2

SHEAR PANELS

2.1 OVERVIEW OF DESIGN PROCEDURE

A flowchart summarizing the design procedure for flat and curved, composite or metal shear panels is shown in Figure 2.1. The various steps involved in the design procedure are detailed in the following paragraphs. The underlying analytical basis for detail design of the shear panels is the generalized tension field theory documented in Reference 1.

The generalized (for application to composites as well as metals) tension field analysis procedures are coded in a computer program called TENWEB that can be used as an efficient design tool. Detailed instructions for the use of this program are given in Reference 2. The equations for analysis incorporated in program TENWEB pertain to cylindrically curved composite panels and to flat composite panels if the radius of curvature in the latter case is set to a very high value (of the order of 10^{10}). Use of appropriate values for the elastic constants in the program permits its direct application to metal panels. In this section, the methodology for accomplishing detail design using TENWEB is demonstrated.

Examples are given to illustrate the application of the methodology.

2.2 DESIGN CRITERIA

The design criteria that need to be established at the outset are:

(a) Materials and material properties,

(b) Design allowable stresses and strains, and

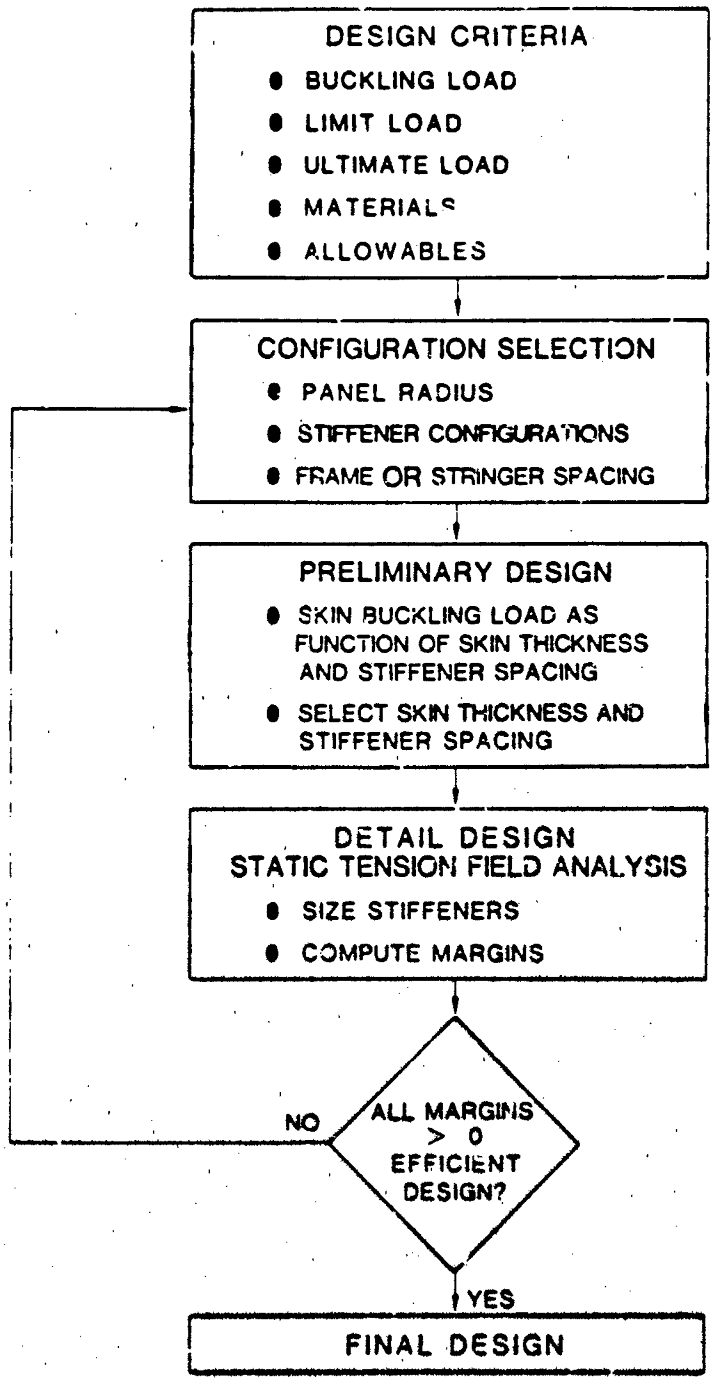


Figure 2.1. Shear Panel Design Procedure

(c) Initial skin buckling load and its relationship to load factor (g-level) and the design limit load.

The material properties that should be established are the elastic constants and the ultimate compression strains (ϵ_{cu}) or stresses (F_{cu}). The latter values are required in the stiffener crippling calculations. The ultimate compression stress values for metals can be obtained from MIL-HDBK-5. For composite materials typical of current usage on military aircraft (e.g., T300/5208, AS/3501-6 graphite epoxies) the ultimate strain ϵ_{cu} can either be determined from unnotched coupon tests or the following values may be used:

$$\epsilon_{cu} = .012 \quad \text{for laminates with at least 40 percent 0-degree plies}$$

$$= .015 \quad \text{otherwise} \tag{1}$$

Design data required for composites are the allowable strains in compression and tension which can be considerably lower than the ultimate values.

The general guideline to be followed in defining the initial buckling load is that the skins must not buckle under loads equivalent to 1-g or less. The 1-g condition corresponds to level flight or ground storage. In order to realize the potential advantages of postbuckled designs, the skin buckling loads must be set between 25 to 35 percent of the design limit load (DLL). Thus, the shear flow at design ultimate load (DUL) ranges between 4 to 6 times the initial skin buckling shear flow. The critical static load conditions provide the basis for defining the design ultimate internal shear flows that the panel must sustain without rupture or collapse.

2.3 CONFIGURATION SELECTION

The overall structural requirements, to a large extent, dictate the selection of a stiffened panel configuration. The size and curvature of the panel are determined by panel location on the actual structure. In many instances the frame spacing is predetermined by the overall structural configuration and, thus, only the stringer spacing needs to be determined in preliminary design. Selection of a stringer spacing and frame spacing is interrelated with the design of the skin for a specified buckling load. These geometric parameters, therefore, are determined in the preliminary design stage.

The most significant decision to be made at this stage is the selection of stringer and frame configurations, i.e., the stiffener cross-sectional shapes. The primary considerations in selecting stiffener cross-sectional shapes are structural efficiency, manufacturing ease, and simplicity of attachment to substructure. The most popular concepts in metal designs have been open-section stiffeners such as I-, J-, Z-, inverted hat and blade sections since they facilitate joints and splices and attachment to substructure. In addition, closed section stiffeners such as hat stiffeners have also been used. In composite panel designs the same stiffening concepts, with the exception of Z-sections, can be used. Z-section stiffeners are not desirable since the single skin attach flange in cocured or adhesively bonded construction does not provide adequate strength under pull-off loads in practical designs.

As a first step in choosing a cross-sectional shape for the stiffeners, a weight comparison of the different concepts for given loading conditions is necessary. Recognizing that the stiffeners in postbuckled shear panels are axial compression load carrying members and that the stiffeners as a whole remain stable up to failure, weight comparisons carried out for stiffened panels under compression loading can be used to evaluate relative efficiencies. Several analytical and experimental studies (e.g., References 3 through 6) have

been conducted to evaluate the relative efficiencies of the commonly used stiffening concepts for metals and composites. The results of Reference 5, in particular, are useful in guiding the selection of stiffener configuration on the basis of weight. These results are summarized in Figure 2.2, reproduced from Reference 5. As is evident from Figure 2.2, the graphite epoxy J- and blade configurations have similar structural efficiencies. However, for graphite-epoxy, the hat section stiffeners provide a 32 percent higher efficiency and, thus, are most desirable in minimizing weight. The trends are similar for metal panels with the hat stiffeners providing a 22 percent efficiency gain as compared to the open section stiffeners. For both material types, the J-section stiffeners have a slight edge in efficiency (approximately 5 percent) over blade stiffeners.

The higher efficiency of hat stiffeners and the ease of manufacturing and attachment of open sections implies that the final stiffener cross-section selection will be a compromise. In general, for curved frame/longeron or curved frame/stringer type construction, hat section stringers and J-section frames provide an efficient combination. For floating frame/stringer type construction used only in metal panels, inverted hat section stringers and J-section frames may be desirable.

2.4 PRELIMINARY DESIGN

The design variables calculated in preliminary design are the skin thickness and the stiffener spacing. The design driver is the skin initial buckling load $N_{xy,cr}$ and the limiting criteria are minimum skin thickness (.04 in. for graphite/epoxy as well as aluminum) and a reasonable stiffener spacing. The design variables to be selected are shown in Figure 2.3 where one bay of the shear panel is shown. The stiffener cross-sectional shapes shown are for reference only.

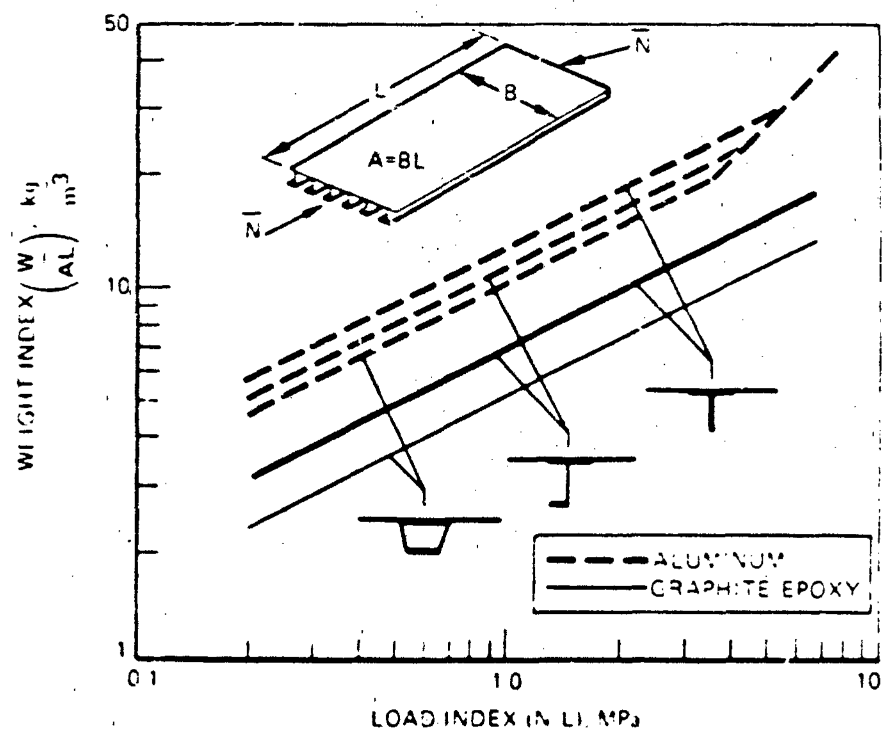


Figure 2.2. Compression Load Structural Efficiency Comparison for Hat-, J-, and Blade Configurations

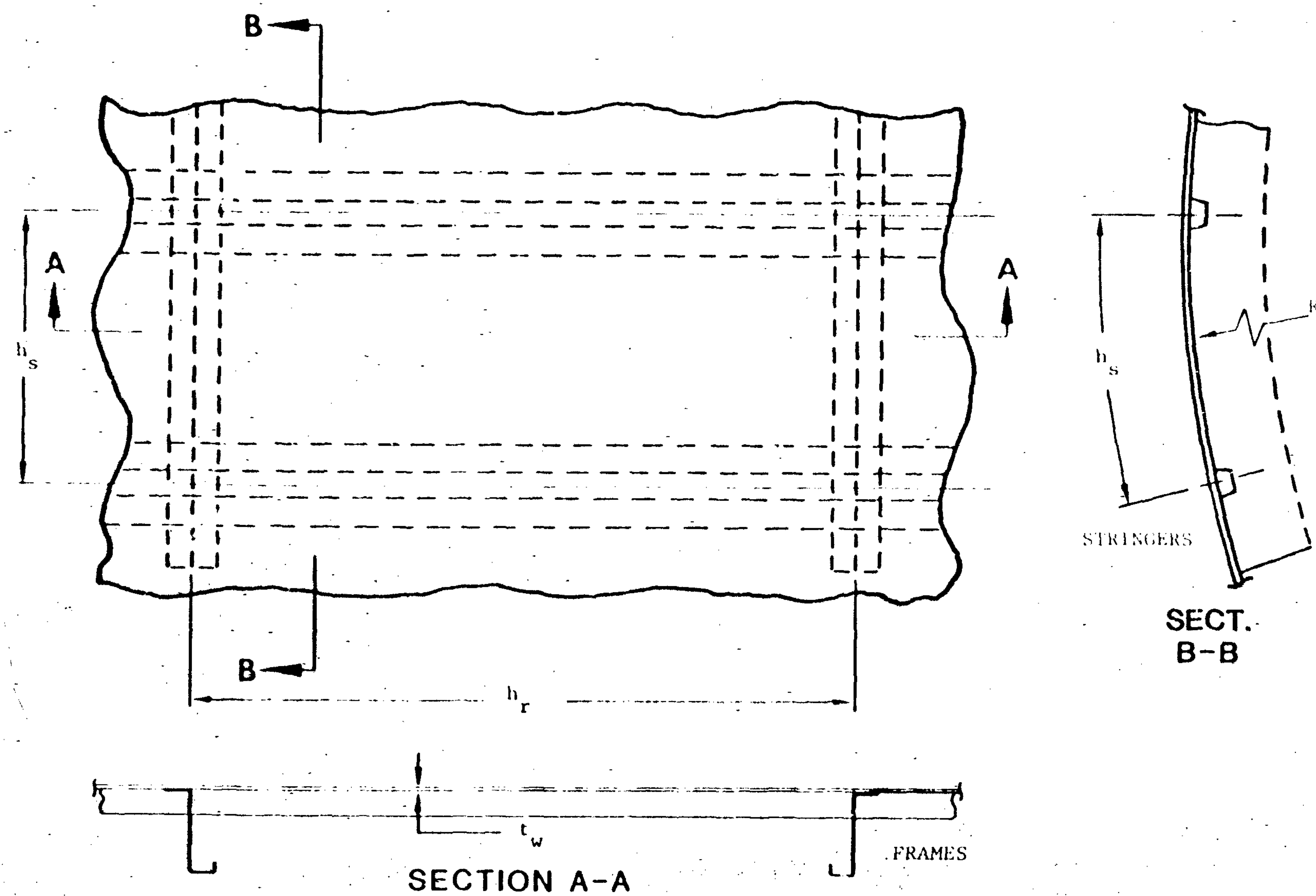


Figure 2.3. Shear Panel Design Variables

The calculations are carried out by first fixing the frame or ring spacing, h_r, and selecting a skin thickness. For composite panels, the number and orientation of plies must also be tentatively selected. If the frame spacing is not predetermined by the overall structural configuration then a value between 15 inches and 30 inches for frame/stringer construction may be selected. For frame/longeron construction, the frame spacing may range between 4 inches to 10 inches.

In order to size the skin, a good starting point is minimum gage thickness dictated by prevalent design practice. The skin thickness may have to be increased in metal panels if countersunk fasteners have to be accommodated. Metal skin mid-bay thicknesses in the range of 0.05 inch to 0.063 inch are most commonly used. Lands milled in the metal skins under stiffeners can serve to accommodate the countersunk fasteners.

Available design data show that for composite panels skin thicknesses slightly greater than the minimum permissible gage are adequate for post-buckled structures. Ply orientations that are predominantly $\pm 45^{\circ}$ are most efficient for buckling critical designs. As in conventional composite construction, the stacking sequence should be balanced and symmetric. Biwoven or unidirectional graphite/epoxy may be used to fabricate the skins. The improved drapability of woven graphite/epoxy facilitates layup of curved skins. Unidirectional 0-degree and 90-degree plies are usually included in the skin layup to resist longitudinal or transverse axial loads or pressure if these loads are present in addition to the shear. Since the 0's and 90's can be used as single plies as opposed to the ±45's which must be used in pairs, the former are also more convenient in building up skin thickness to a specific requirement.

On the basis of above consideration, if only the shear buckling criteria has to be met then layups such as $[\pm 45]_{2S}$, $[\,\underline{45}\,]_4$, $[\,\underline{45}_2/0/\underline{45}_2]$, $[\,\underline{45}/90/0/90/\underline{45}\,]$, where $\underline{\quad}$ denotes a woven ply, may be initially selected for

the skin laminate. Extra plies may be added during the course of the design iteration.

Shear Buckling of Skin

The next step consists in calculating the skin buckling load $N_{xy,cr}$ as a function of the stringer spacing h_s. These calculations have to be carried out for each skin thickness being considered and in the case of composites for each ply layup. The shear buckling stress for composite skins can be calculated using program SS8 documented in Reference 7. The skin boundary conditions are assumed to be fixed at the curved frames and at the stringers. The curved metal panel initial buckling stress can be calculated using the following equation:

$$\left.\begin{aligned} \tau_{cr,\ elastic} &= \frac{K_{s1}\pi^2 E h_s^2}{12R^2Z^2} && \text{if } h_r > h_s \\ &= \frac{K_{s2}\pi^2 E h_r^2}{12R^2Z^2} && \text{if } h_s > h_r \end{aligned}\right\} \quad (2)$$

where,

K_{s1}, K_{s2} = critical shear stress coefficients for simply supported curved plates determined from Figures 2.4 and 2.5 (Reference 8).

For flat metal skins the elastic buckling stress is determined using the following equation:

$$\tau_{cr} = K_s E_c \left(\frac{t}{h_s}\right)^2 \quad (3)$$

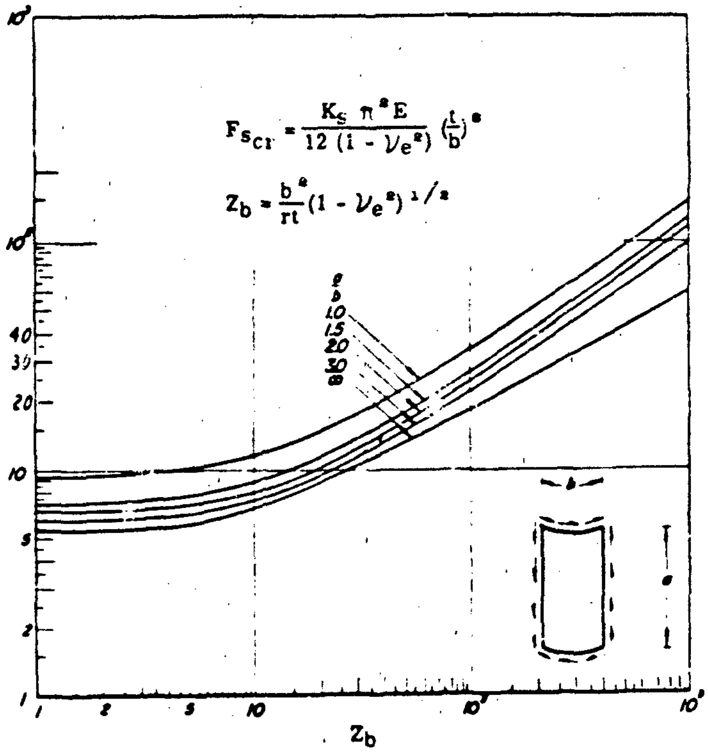


Figure 2.4. Shear Buckling Coefficient for Simply Supported Curved Metal Panel. Curved Edge Shorter than Straight Edge

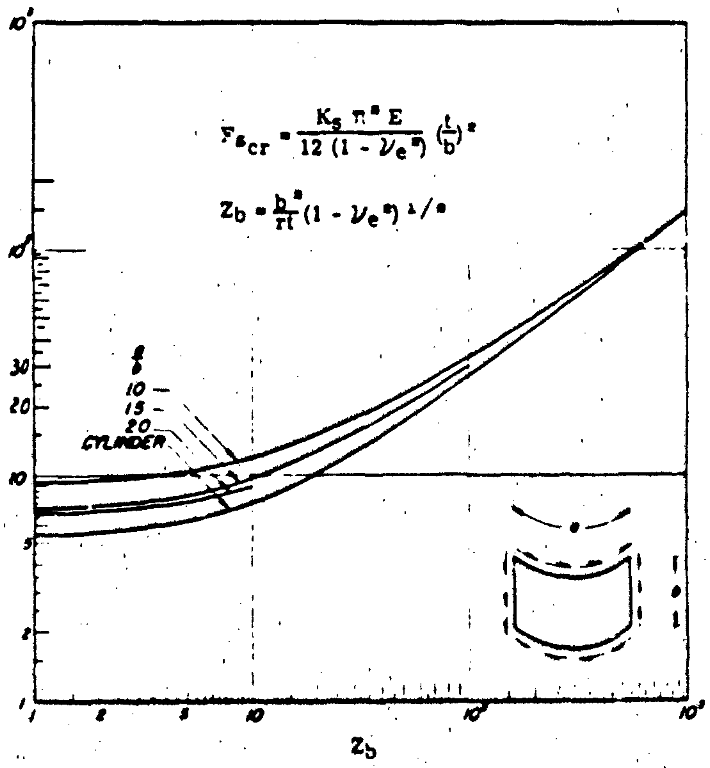


Figure 2.5. Shear Buckling Coefficient for Simply Supported Curved Metal Panel. Curved Edge Longer than Straight Edge

with

$$K_s = 4.83 + 3.61 \left(\frac{h_s}{h_r} \right)^2$$

which is plotted in Figure 2.6.

The metal panel skins in both cases are assumed to be simply supported at the stringers and the frames.

In Equations 3 and 4,

h_s is stringer spacing, in

h_r is ring or frame spacing, in

R is panel radius, in

t is skin thickness, in

E_c is the compression modulus of the skin, in

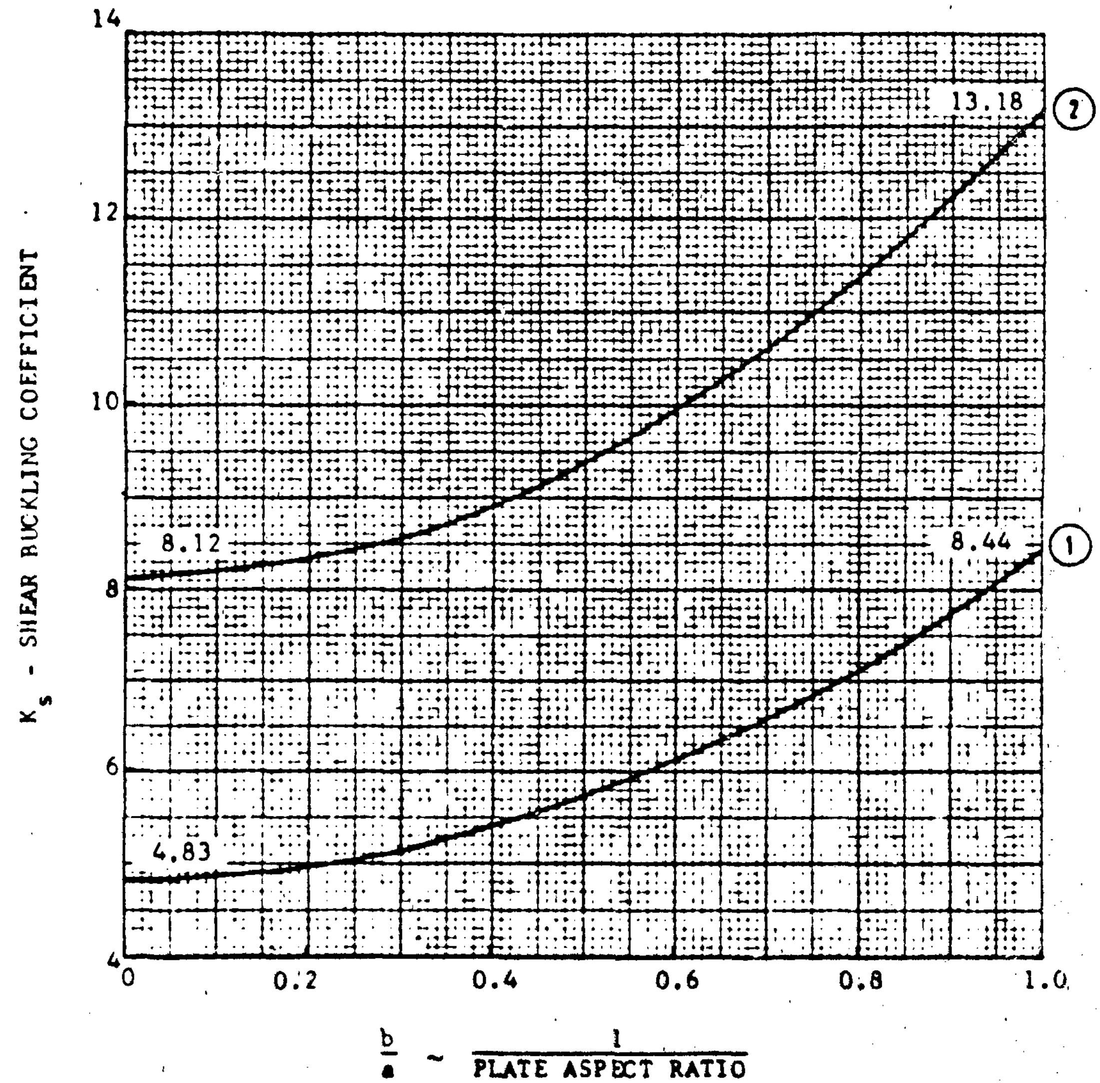
$$Z = \frac{h_s^2}{Rt_w} \sqrt{(1 - \nu^2)} \qquad \text{if } h_r > h_s$$

$$= \frac{h_r^2}{Rt_w} \sqrt{(1 - \nu^2)} \qquad \text{if } h_s > h_r$$

ν is the Poisson's ratio for the skin material

Selection of Skin Thickness and Stringer Spacing

The skin thickness and stringer spacing are selected from plots of the calculated buckling loads versus the stringer spacing. In order to illustrate the procedure, two such plots corresponding to Design Example No. 1 at the end of this section are shown in Figure 2.7. Referring to the figure, a buckling parameter λ, equal to the ratio of the calculated buckling load and the design buckling load, is plotted against the stringer spacing h_s. The buckling loads were calculated assuming clamped boundary conditions at the



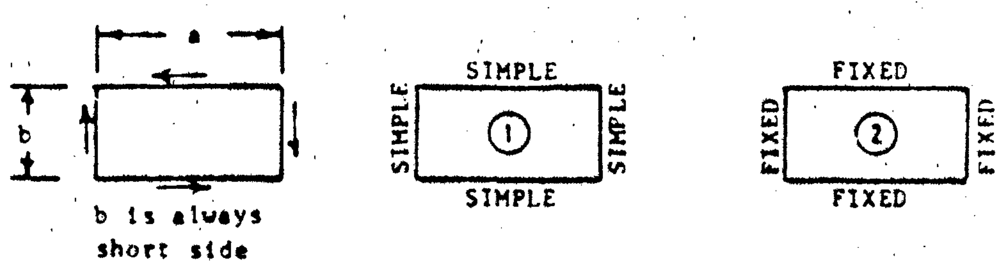


Figure 2.6. Buckling Coefficient for Flat Shear Parels

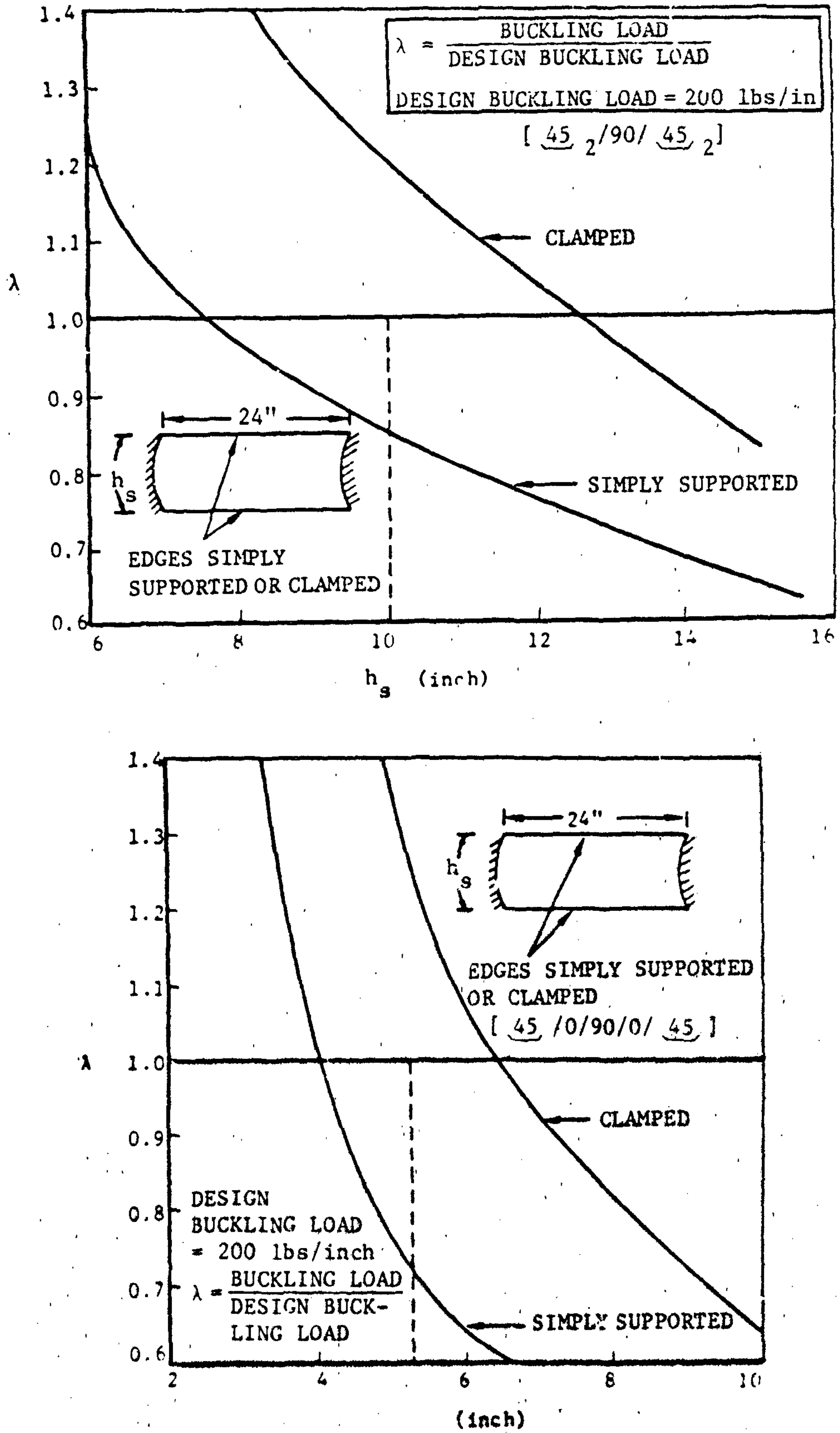


Figure 2.7. Buckling Load Versus Stringer Spacing for [45 2/90/ 45 2] and [45 /0/90/0/ 45] Skin Layups

frames and a fixity intermediate between simply supported and clamped at the stringers. As is evident from Figure 2.7, the [$\underline{45}_2$/0/ $\underline{45}_2$] skin layup with a 10-inch stringer spacing is the preferred design since for the thinner skin with a [$\underline{45}$ /90/0/90/ $\underline{45}$] layup the narrower stringer spacing is bound to impose a weight penalty. Thus, a selection of skin thickness and stringer spacing can be made by comparison of such plots for the various skin thicknesses and layups that were initially picked for evaluation.

2.5 DETAIL DESIGN

Detail design of shear panels involves sizing of the stringers and the frames and computing margins for the various possible failure modes. The procedure is iterative in that initial sizes are assumed for the stiffeners, the margins computed, and if any of the margins are negative or too high, the stiffeners are resized and new margins are computed. This iteration is continued till all margins are positive and reasonable in magnitude so that any weight penalties are minimized. The various steps in the detail design procedure are described in the following paragraphs.

Initial Stringer and Frame Dimensions

The stringer and frame cross-sectional shapes were selected in Section 2.3. For metal panels, the initial dimensions are determined by selecting a standard section such as the AND-series I, J or Z sections. The stiffener cross-sectional area selected for the first iteration may be arbitrary unless historical data are available or geometric constraints dictate certain dimensions. Exact section dimensions can be determined only after several iterations.

In the case of composite panels, on the basis of structural efficiency, the most commonly used stiffener shapes are hat, J or blade sections.

The selection of initial stiffener sizes in this case requires a definition of the ply composition for various elements of the stiffener in addition to the dimensions. Studies on optimizing stiffener cross-sections conducted in References 3 and 5 have led to the general guidelines shown in Figure 2.8 for selecting efficient and practical layups in the design of stiffeners under axial compression loads. The recommended additional 0-degree plies in the skin should be utilized to ensure a slight taper in the stiffener flange bonded or cocured to the skin. This can be accomplished by gradually dropping-off the 0-degree plies as shown in Figure 2.9. The smooth transition from the stiffener flange to the skin is essential for stiffener/skin interface strength.

The composite stiffener dimensions that need to be selected are shown in Figure 2.9 as the widths b_i and the thicknesses t_i. For initial sizing, typical range of values for the stiffener element widths and the ply distribution are shown in Figure 2.10. These dimensions were obtained from a survey of panel designs that have been tested and must be treated as guidelines only.

Effective Stiffener Areas

Calculation of effective stiffener areas must take into account the presence of lands in metal skins and ply drop-offs in composite skins. In metal skins if a web land occurs in conjunction with the stiffener, the increase in web thickness is assumed an integral part of the stiffener. For composite panels the thickness of stiffener flanges attached to the skin is defined as the average thickness of the tapered flange-skin combination with the width equal to the actual flange width. The skin under the cap of a hat section stiffener is assumed to be an integral par of the stiffener.

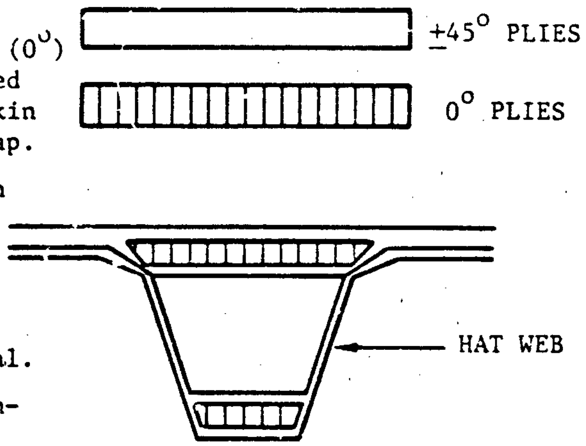
Hat Section Stiffeners

1. High axial stiffness (0°) plies should be placed in the hat cap and skin directly above the cap.

   Reason: Provide high bending stiffness to resist overall buckling of the panel.

2. Hat webs should be entirely $\pm 45^\circ$ material.

   Reason: Minimize compression load in web and provide increased shear stiffness.



J and Blade Section Stiffeners

1. High axial stiffness plies in cap and in skin under stiffener.

   Reason: High bending stiffness stiffener.

2. Stiffener webs should be entirely $\pm 45^\circ$ material.

   Reason: Minimize axial load in webs, thus, suppressing local buckling.

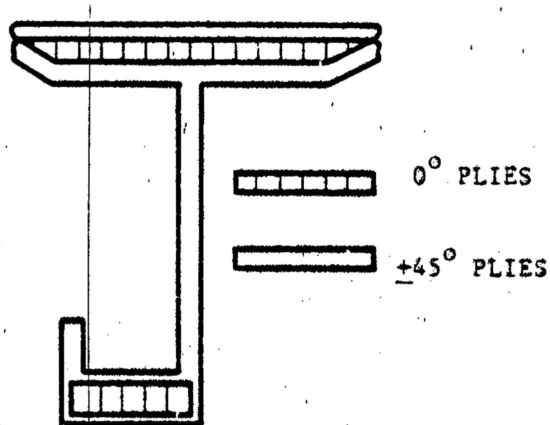


Figure 2.8. General Guidelines for Selecting Ply Distribution in Stiffeners Under Axial Compression

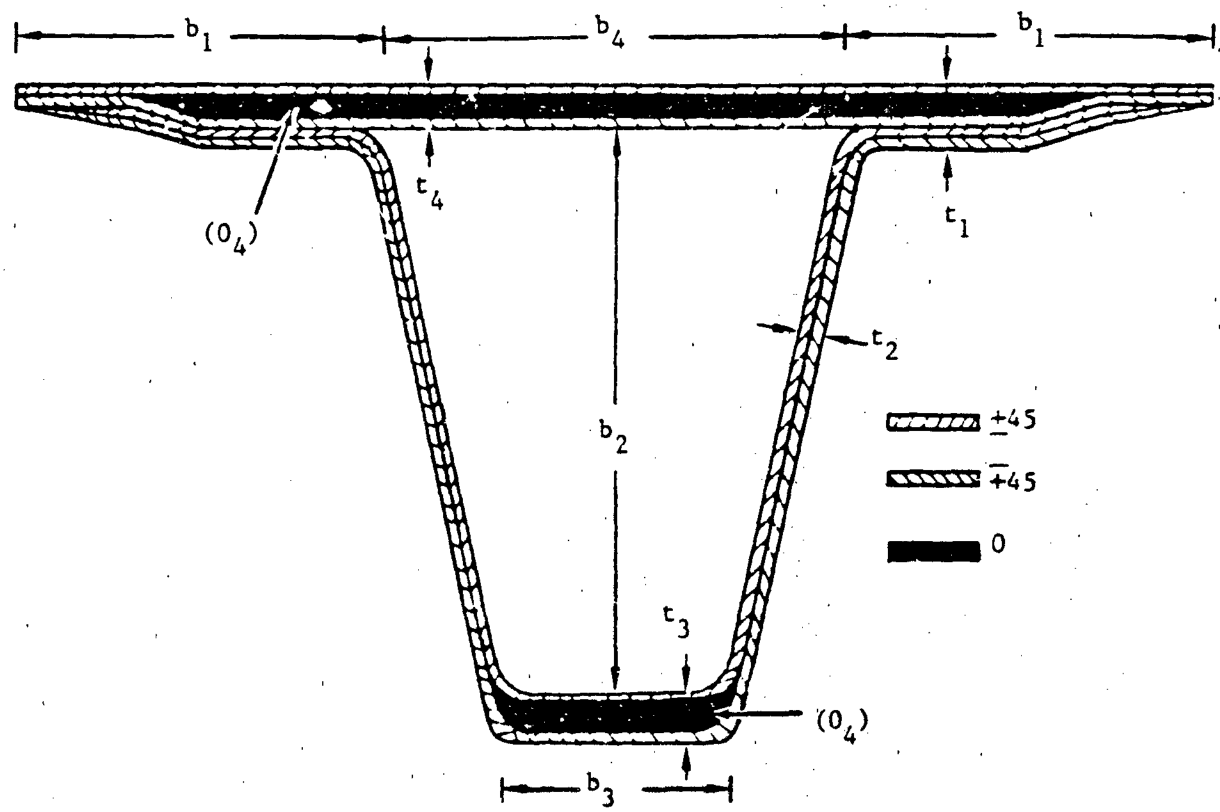
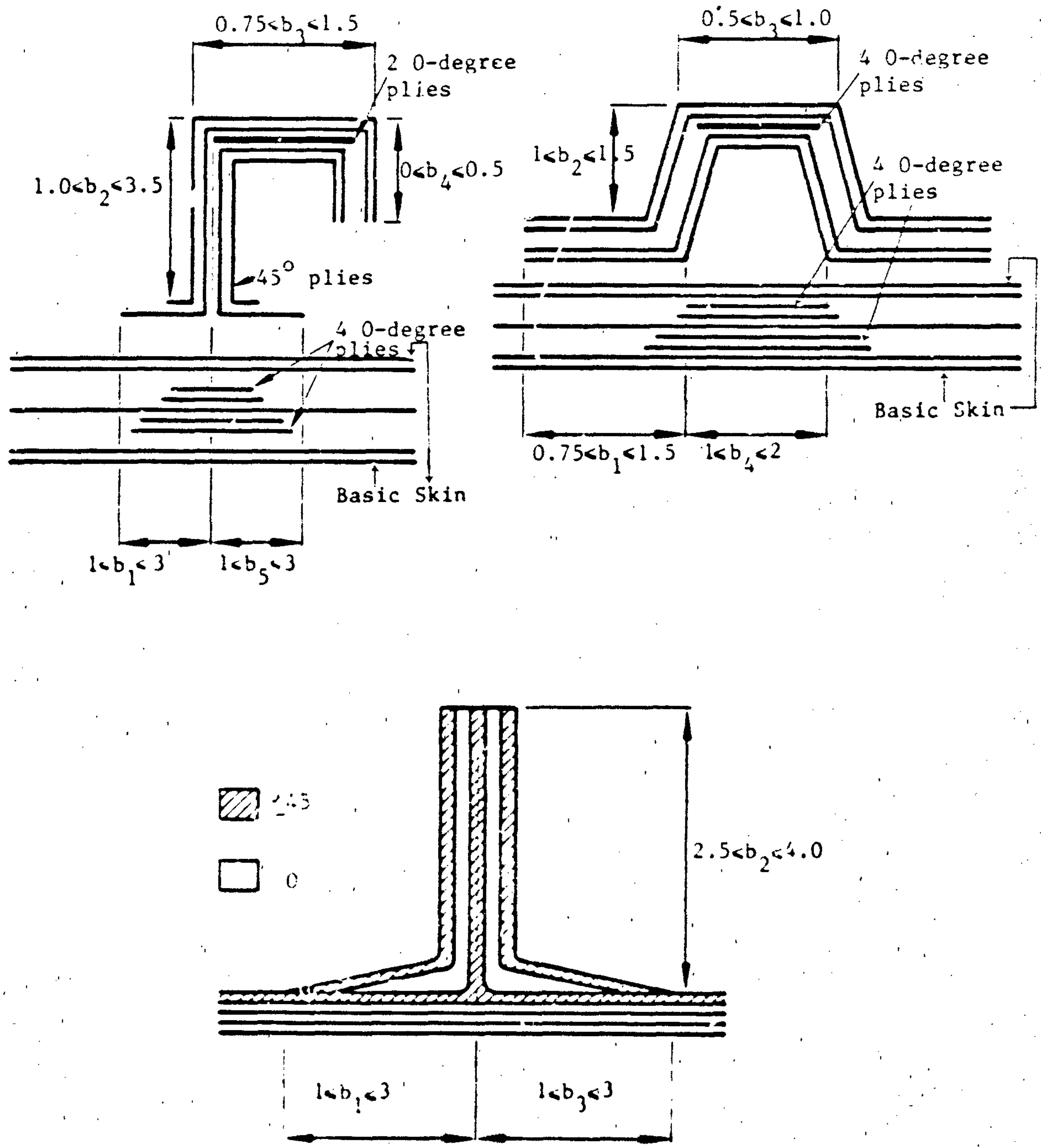


Figure 2.9. Ply Drop-Offs in Hat Section Stiffener and Stiffener Design Variables



ALL DIMENSIONS IN INCHES

Figure 2.10. Typical Stiffener Dimensions for Initial Sizing

Stiffener Sizing and Margin Computation

This step is the crux of the detail design activity. Stiffener sizing and margin computation for shear panels is accomplished using the static tension field analysis of Reference 9 which was modified to account for material anisotropy in Reference 1. The basic semiempirical equations in the analysis and the failure criteria are detailed in Reference 1. These equations including the failure analysis for modes common to metal and composite panels have been coded in computer program TENWEB which is documented in Reference 2. The semiempirical equations are not repeated here. The emphasis instead is on demonstrating the use of TENWEB in designing postbuckled panels. Failure modes that are unique to metals or composites and which have to be checked for manually are given in this Guide.

Shear Panel Failure Modes

The possible failure modes that have to be checked for in designing shear panels are:

(a) Column stability of stringers and rings or frames,

(b) Stability of the entire panel,

(c) Forced crippling of stringers and frames,

(d) Stiffener/skin separation for composite panels,

(e) Permanent set in metal skins due to yielding in diagonal tension, and

(f) Skin rupture in metal and composite panels.

- Ultimate failure in shear for metals
- Diagonal tension failure in composites

Checks for failure modes A through D are incorporated in program TENWEB. The last two modes have to be checked for manually.

Tension Field Analysis (TENWEB)

The essential elements of the generalized (for application to metals and composites) tension field theory as coded in program TENWEB are shown in Figure 2.11. The diagonal tension factor k characterizes the degree to which diagonal tension is developed in the skin of stiffened panels loaded in shear. A value of $k = 0$ characterizes an unbuckled skin with no diagonal tension; a value of $k = 1.0$ characterizes a web in pure diagonal tension. The diagonal tension factor is computed using the following expression:

$$K = \mathrm{Tanh}\left[\left(0.5 + 300\,\frac{t_w h_r}{R\, h_s}\right)\log\frac{\tau}{\tau_{cr}}\right]$$

where,

t_w = web thickness

h_r = ring spacing

h_s = stringer spacing

R = panel radius

τ_{ult} = design ultimate shear stress

τ_{cr} = buckling shear stress of web

The stiffener sizing commences by first computing the diagonal tension factor for the stringer and ring spacing skin thickness, and panel

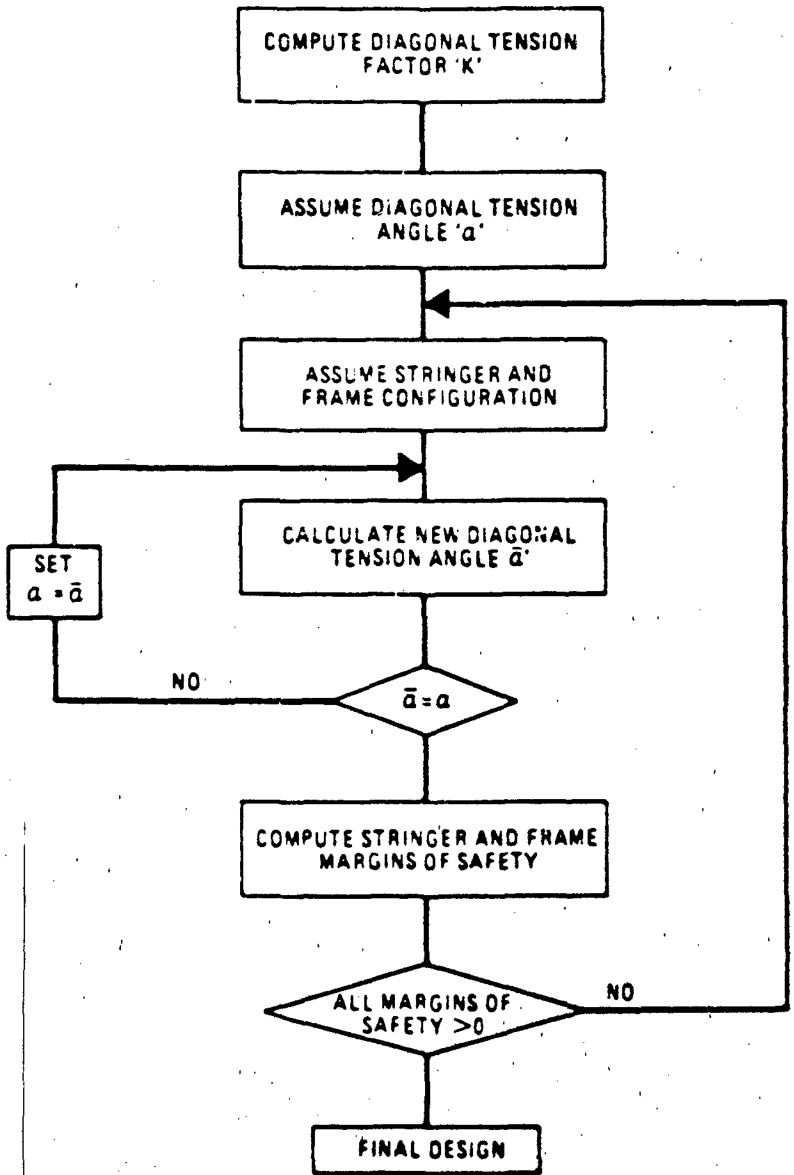


Figure 2.11. Application of Tension Field Theory to Shear Panels.

radius selected in the preliminary design stage. The assumed initial stringer and ring dimensions are entered in TENWEB and the margins for failure modes A through D given in the preceeding subsection are computed. If the margins are not positive then the stiffeners have to be resized.

In most practical cases, i.e., if the initial dimensions of the stiffener are selected based on the guidelines given in Figure 2.10, the design driver will be the forced crippling mode of failure for stringers and/or rings. The other stability requirements are, in general, easily met. The approach to resizing stiffeners for positive margins on forced crippling is defined by examining the equations relating to this mode of failure:

Stringer forced crippling strain:

$$\varepsilon_{os} = 0.00058 \left[\left(\frac{\varepsilon_{all} E_{cs}}{1000} \right)^{0.4} k^{2/3} \left(\frac{t_{us}}{t_w} \right)^{1/3} \right] \tag{5}$$

Ring or frame forced crippling strain:

$$\varepsilon_{or} = 0.00058 \left[\left(\frac{\varepsilon_{all} E_{cr}}{1000} \right)^{0.4} k^{2/3} \left(\frac{t_{ur}}{t_w} \right)^{1/3} \right] \tag{6}$$

where ϵ_{all} is the laminate allowable strain, E_{cs} and E_{cr} are the modulus of the stringer and ring leg attached to the web, respectively, and t_{us} and t_{ur} are the thickness of the stringer and the ring leg attached to the web.

Based on Equations 5 and 6 stiffener resizing should proceed as follows:

(a) Increase t_{us} and/or t_{ur}, i.e., increase thickness of stiffener leg attached to the web,

(b) In the case of metal panels select new stiffener material with a higher yield strain and compression modulus, and

(c) For composite stiffeners ϵ_{all} is generally in the order of 0.004 to 0.006 and cannot be varied significantly. However, the compression modulus depends on the distribution of plies. Thus, tailoring of the stiffener cap and the skin below the stiffener by increasing the number of 0° plies can be used to increase the compression modulus.

The new stiffener size, thus, selected is entered in TENWEB and new margins are computed. This procedure is repeated till a desired set of margins have been achieved. The process is very quick and at most two or three iterations can lead to final stiffener sizes.

The margins are computed as follows:

$$\text{Margin of Safety} = \frac{\epsilon_{os} \text{ (or } \epsilon_{or})}{\epsilon_{smax} \text{ (or } \epsilon_{rmax})} - 1$$

where, ϵ_{smax} and ϵ_{rmax} are the maximum stringer and ring strains, respectively.

Skin Rupture and Permanent Set Checks

The stiffened panel design obtained from TENWEB is now checked for skin rupture in the case of metal and composite panels and for permanent set in the case of metal panels. These checks are performed as follows:

(a) Metal Skins:

The ultimate allowable shear stress in metal skins is given by:

$$F_s = 0.9\, F_{ty} \left[1 + 0.5 \left(\frac{F_{tu}}{F_{ty}} - 1 \right)^2 \right] \left[0.5 + (1-k)^3 \left(\frac{F_{su}}{F_{tu}} - 0.5 \right) \right] \quad (7)$$

where,

F_s is the ultimate allowable web shear stress, psi.

F_{tu} is the allowable ultimate tension stress for the web material, psi.

F_{ty} is the allowable tension yield stress for the web material, psi.

F_{su} is the allowable ultimate shear stress for the web material, psi.

Equation 7 is limited to essentially isotropic metallic materials. In cases where a slight difference exists in the mechanical properties in the longitudinal (L) and long transverse (LT) directions, use the minimum properties.

Since the equation was obtained by a fit to test data, the effects of plasticity are included.

In general, permanent set in the skin at limit load is not permitted. The maximum allowable value of the diagonal tension factor at ultimate shear stress (k_{all}) to prevent permanent buckling of the skin at limit load is given by:

$$k_{all} = 0.78 - (t - 0.012)^{0.5} \tag{8}$$

This equation is based on flat aluminum metal panel data and is conservative for curved panels.

(b) Composite Skins:

TENWEB provides as output the diagonal tension strain in the skin at design ultimate shear flow (ϵ_{DT}). In order to prevent skin rupture, therefore,

$$\epsilon_{DT} < \epsilon_{tu} \tag{9}$$

where, ϵ_{tu} is the allowable tensile strain for the material.

Final Design

If for the last two manual checks the margins are found to be negative then the skin thickness has to be increased and in the case of composite panels the ply composition altered. This necessitates returning to the preliminary design stage and then repeating detail design.

2.6 EXAMPLES

The shear panel design procedure outlined above is demonstrated by way of the following composite and metal examples.

Example 1. Curved Composite Panel

A postbuckled composite shear panel with a 45 inch radius is to be designed to carry a design ultimate shear flow of 900 lbs/in with adequate margins. Skins are not permitted to buckle below 33 percent of design limit load. The frame spacing $h_r = 24$ in.

Design Procedure:

(a) Design criteria:

The materials selected are:

AS/3501-6 unidirectional graphite/epoxy for reinforcement of stiffener caps and skin under stiffener.

A370-5H/3501-6 woven graphite/epoxy for skins and stiffeners.

Lamina Properties:

	A370-5H/3501-6	AS/3501-6
Per ply thickness, in	.013	.0052
EL, psi	10.0×10^6	18.7×10^6
ET, psi	9.2×10^6	1.87×10^6
GLT, psi	0.9×10^6	0.85×10^6
NULT	0.055	0.3

Material Allowables:

ε_{all} = 0.004 in tension and compression

Loads:

Design ultimate shear flow (DUL) = 900 lbs/in

Design limit shear flow (DLL) = 600 lbs/in

Initial skin buckling load (IBL) = 200 lbs/in

(b) Configuration Selection:

$$\left.\begin{array}{l}\text{Panel radius } R = 45 \text{ in} \\ \\ \text{Frame spacing } h_r = 24 \text{ in}\end{array}\right\} \text{Given}$$

Skins to be designed primarily for buckling.

Select viable skin layups:

Layup 1 - [$\underline{45}_2$/90/$\underline{45}_2$] underscore denotes a woven ply

t = .0572 in

Layup 2 - [$\underline{45}$/90/0/90/$\underline{45}$]

t = .0416 in

Select stiffener cross-sectional shape on the basis of efficiency and ease of attachment to substructure

- Hat section stringers selected for efficiency
- J section frames selected for efficiency and ease of attachment to substructure.

(c) Preliminary Design:

(1) Obtain skin buckling load ($N_{xy,cr}$) as a function of stringer spacing (h_s) using program SS8 for fixed and simply supported boundary conditions at the stringers and fixed boundary conditions at the frames. Both layups to be considered.

(2) $N_{xy,cr}$ versus h_s plots for the two layups are shown in Figure 2.7.

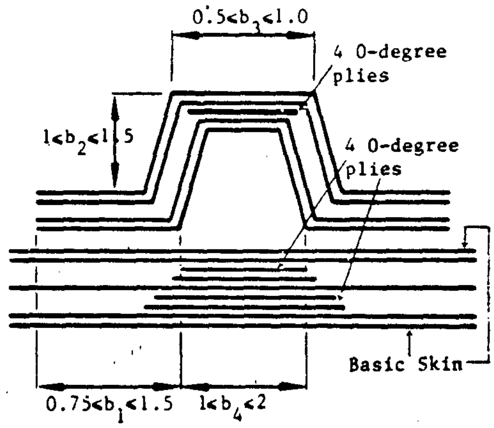
(3) Skin layup 1 with $[\pm 45_2/0/\pm 45_2]$ orientation of plies with larger stiffener spacing selected for efficiency and reduced manufacturing cost.

(4) h_s = 10 in., t = .0572 in.

(d) Detail Design:

(1) Select initial dimensions and ply distribution for stiffeners using the range of values given in Figure 2.10 and previous experience.

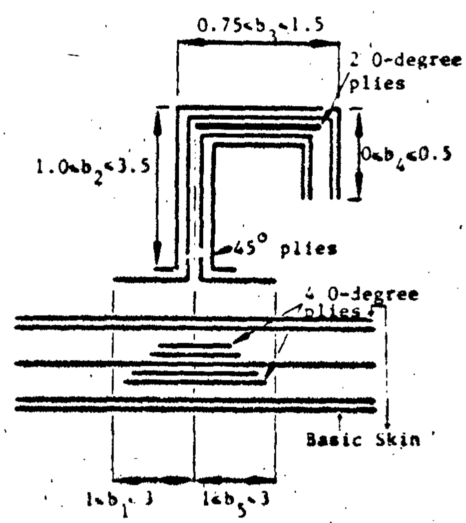
o Hat Section Stringer:



Initial Dimensions and Layup for Hat Section Stringer:

ELEMENT NO.	WIDTH, IN.	LAYUP
1	$b_1=1.0$	$[\underline{45}_6/90/0_2/\underline{45}_2]$
2	$b_2=1.3$	$[\underline{45}]_4$
3	$b_3=0.75$	$[\underline{45}_2/0_7/\underline{45}_2]$
4	$b_4=1.2$	$[\underline{45}_2/0_3/90/0_3/\underline{45}_2]$

o J-section frame:



Initial Dimensions and Layup for J-section frame:

ELEMENT NO.	WIDTH, IN.	LAYUP
1	b_1=1.0	$[\underline{45}_4/90/0/\underline{45}_2]$
2	b_2=0.5	$[\underline{45}_4/90/0/\underline{45}_2]$
3	b_3=2.9	$[\underline{45}]_4$
4	b_4=1.0	$[\underline{45}_2/0_2/\underline{45}_2]$
5	b_5=0.4	$[\underline{45}]_4$
6	b_6=0.5	$[\underline{45}_4/90/0/\underline{45}_2]$
7	b_7=1.0	$[\underline{45}_4/90/0/\underline{45}_2]$

(2) Run program TENWEB for margin computations

TENWEB output for initial stiffener dimensions

PROPERTY	STRINGER	FRAME	SKIN
EA, lbs	0.31×10^7	0.22×10^7	
EI, lbs-in^2	0.9×10^6	0.33×10^7	
t_u*, in	0.112	0.088	
y, in ***	0.346	0.81	
Crippling Strain	0.0019	0.0017	
Maximum Strain	-0.0024	-0.0029	.003029**
Margin, %	-20	-40	32
All other margins	OK	OK	OK

* t_u equals thickness of stiffener leg attached to skin

** Diagonal tension strain

Margin based on ϵ_{tu} = .004

*** Location of neutral axis with respect to skin midsurface.

The margins on stringer and frame forced crippling are both negative. Therefore, the stiffeners must be resized.

(3) Consider first resizing of the hat stiffener. In order to increase the stringer crippling strain the following actions are taken:

(a) Increase the width and thickness t_u of element 1 by changing b_1 from 1.0 to 1.2 in. and increase the number of 0-degree plies in the skin under the stiffener from 6 to 8. Of the two additional plies in the skin under the stiffener only one is fully effective with element 1 due to ply drop-offs.

(b) Increase the compression modulus of stringer by adding 2 more 0-degree plies in the cap.

(c) Increase the width of element 4 from 1.2 to 1.5 in.

The resized stringer properties are:

ELEMENT NO.	WIDTH, IN.	LAYUP
1	b_1=1.2*	$[\underline{45}_6/90/0_3/\underline{45}_2]$*
2	b_2=1.3	$[\underline{45}]_4$
3	b_3=0.75	$[\underline{45}_2/0_9/\underline{45}_2]$*
4	b_4=1.5*	$[\underline{45}_2/0_4/90/0_4/\underline{45}_2]$*

*Denotes changes from initial size

TENWER output for the 2nd iteration:

PROPERTY	STRINGER	FRAME	SKIN
EA, lbs	0.43×10^{7}	0.32×10^{7}	
EI, $lbs\text{-}in^2$	0.11×10^{6}	0.45×10^{7}	
t_u*, in	0.125	0.094	
y, in	0.29	0.774	
Crippling Strain	0.0021	0.0020	
Maximum Strain	-0.0018	-0.0030	.003091
Margin, %	11	-33	29
All other margins	OK	OK	OK

The stringer margin is now positive and the frame margin has improved although still negative. Hence, another iteration is required.

(4) Resize the J-section frame as follows:

(a) Increase the thickness of elements 1 and 2 by increasing the number of 0-degree plies in the skin under the J-section. Add 4 more 0-degree plies to the skin under the stiffener. This effectively adds 3 extra 0-degree plies in elements 1, 2, 6 and 7.

(b) Add 4 more 0-degree plies in the frame cap to increase the axial modulus of the section.

The resized J-section properties are:

ELEMENT NO.	WIDTH, IN.	LAYUP
1	b_1=1.0	[$\underline{45}$ $_4$/0_2/90/0_2/ $\underline{45}$ $_2$]*
2	b_2=0.5	[$\underline{45}$ $_4$/0_2/90/0_2/ $\underline{45}$ $_2$]*
3	b_3=2.9	[$\underline{45}$]$_4$
4	b_4=1.0	[$\underline{45}$ $_2$/0_6/ $\underline{45}$ $_2$]*
5	b_5=0.4	[$\underline{45}$]$_4$
6	b_6=0.5	[$\underline{45}$ $_4$/0_2/90/0_2/ $\underline{45}$ $_2$]*
7	b_7=1.0	[$\underline{45}$ $_4$/0_2/90/0_2/ $\underline{45}$ $_2$]*

* Denotes changes from initial size

TENWEB output for the 3rd iteration:

PROPERTY	STRINGER	FRAME	SKIN
EA, lbs	0.43×10^7	0.93×10^7	
EI, lbs-in^2	0.11×10^6	0.95×10^7	
t_u*, in	0.125	0.104	
y, in	0.29	0.323	
Crippling Strain	0.0021	0.0021	
Maximum Strain	-0.0017	-0.0012	.002997
Margin, %	25	70	33
All other margins	OK	OK	OK

All margins positive. The predicted failure mode is stringer forced crippling.

Example 2

A postbuckled aluminum shear panel of 45 in. radius is to be designed to carry a design ultimate shear flow of 900 lbs/in. The stringer

spacing is to be 10 in. and the frame spacing equals 24 in. The skins are not permitted to buckle below 33 percent of the design limit load. The stringers and frames are Z-sections.

<u>Design Procedure</u>:

(a) Design Criteria:

Material: 7075-T6 aluminum

Properties: $E = 10.6 \times 10^6$, $\nu = 0.3$

Yield Strain: .0061

Loads:

Design ultimate shear flow, N_{xy} (DUL) = 900 lbs/in

Design limit shear flow (DLL) = 600 lbs/in

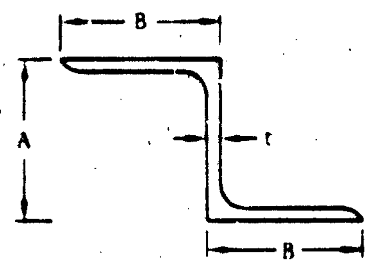
Initial buckling load (IBL) = 200 lbs/in

(b) Detail Design:

(1) Calculate skin thickness to meet IBL requirement. From Figure 2.4, $t_w = 0.063$ in.

(2) Select standard angles for stringers and frames.

DIMENSION	STRINGER	FRAME
A	1.25	1.375
B	0.75	0.75
t	0.063	0.063
STANDARD DESIGNATION	AND10138-1201	AND10138-1301



(3) Run program TENWEB for margin computations.

TENWEB output for stiffener initial dimensions:

PROPERTY	STRINGER	FRAME	SKIN
EA, lbs	0.173×10^7	0.181×10^7	
EI, lbs-in^2	0.472×10^6	0.583^7	
y, in	0.625	0.687	
Crippling Strain	0.0026	0.0026	
Maximum Strain	-0.0059	-0.0062	.0027
Margin, %	-56	-58	126
All other margins	OK	OK	OK

(4) Select heavier angles for stringers and frames:

AND10138-1306 for stringer
AND10138-1306 for frame
} A=1.375, t=.125, B=1.25

(5) Run program TENWEB for margin computations:

PROPERTY	STRINGER	FRAME	SKIN
EA, lbs	4.8×10^7	0.171×10^6	
EI, lbs-i ²	1.45×10^6	1.45^6	
y, in	0.687	0.687	
Crippling Strain	0.0037	0.0037	
Maximum Strain	-0.0026	-0.0030	.002286
Margin, %	40	23	167
All other margins	OK	OK	OK

Thus, the panel will fail by frame crippling at a load 23 percent higher than DUL.

SECTION 3

COMPRESSION PANELS

3.1 OVERVIEW OF DESIGN PROCEDURE

A flow chart summarizing the design procedure for composite or metal compression panels is shown in Figure 3.1. The various steps involved in the design procedure are detailed in the following paragraphs. The design methodology for compression panels is semiempirical and the basic equations used are documented in Reference 1. Application of the methodology to metal panels is straight forward and, therefore, easily implemented by hand calculation. In the case of composite panels, since the constitutive relations are layup dependent, a computer program called CRIP (Reference 2) was written to facilitate the design. Detailed instructions for the use of this program are given in Reference 2. Program CRIP includes an approximate closed form expression to calculate the initial skin buckling strain for curved composite panels. A more accurate buckling strain calculation can be carried out using program SS8 (Reference 7). The failure analysis in CRIP is applicable to both flat and curved skins. In this section the methodology for accomplishing detail design of composite compression panels using CRIP is outlined.

Examples of metal and composite panel designs are given to illustrate the procedure.

3.2 DESIGN CRITERIA

As in the case of shear panels, the design criteria that need to be established at the outset are:

(a) Materials and material properties.
(b) Design allowable stresses and strains, and

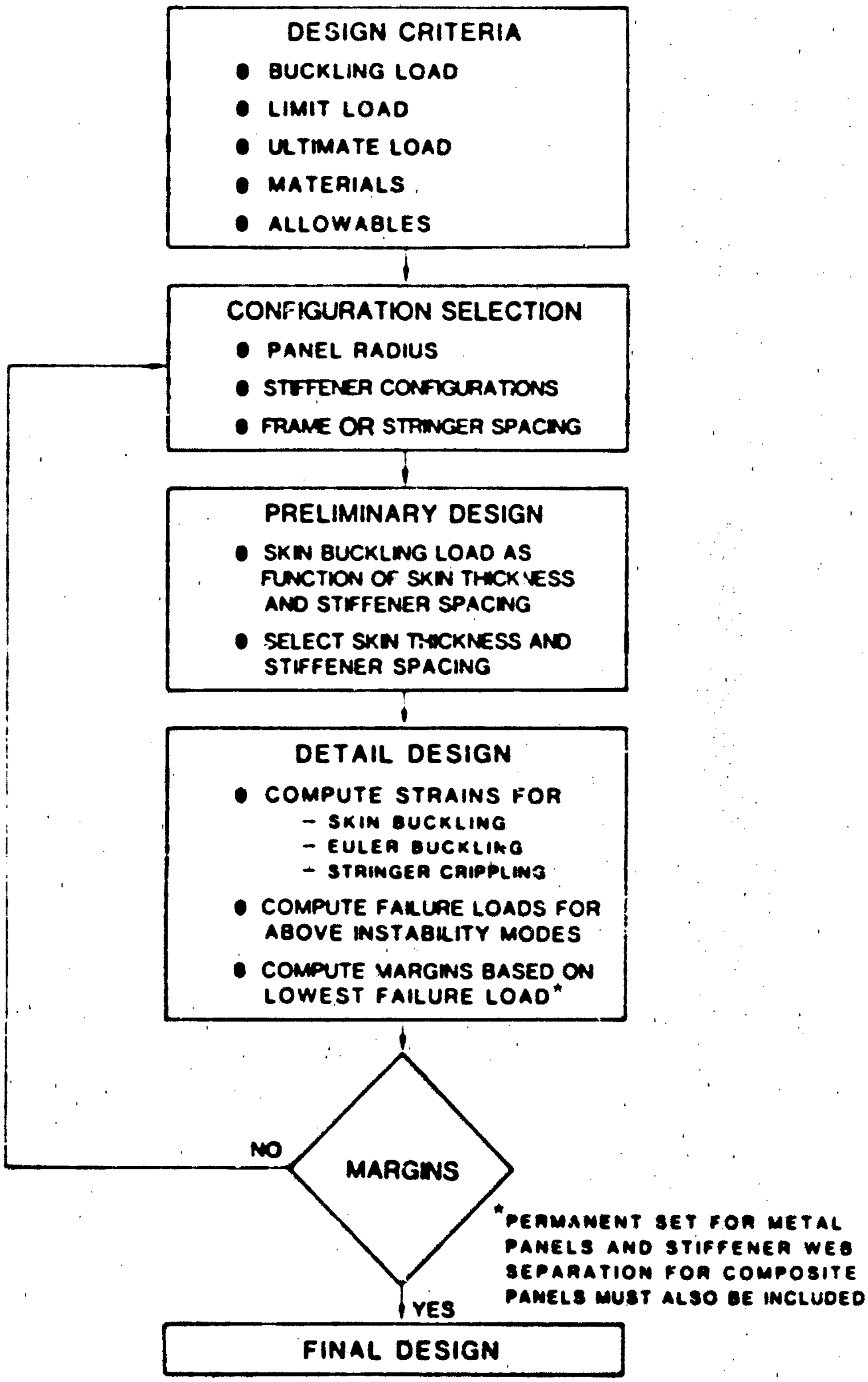


Figure 3.1. Compression Panel Design Procedure

(c) Initial skin buckling load and its relationship to the load factor (g-level) and the design limit load.

The material properties and design allowable data required are identical to those listed in Section 2.2.

The skins are not permitted to buckle at or below 1-g load. In general, panels should be designed so that skin buckling occurs between 25 and 35 percent of the design limit load and the panel is able to sustain design ultimate load without failure or collapse.

3.3 CONFIGURATION SELECTION

In this step, the overall panel geometry, panel curvature and the configuration of stiffeners have to be selected. For uniaxial compression loading the stringers are the primary load carrying members. The frame spacing defines the effective skin length for buckling calculations. The key decision in this step is the selection of a stiffener configuration and a series of skin thicknesses for evaluation.

The structural efficiency comparison of Figure 2.2 together with manufacturing ease and substructure attachment considerations are used in selecting the stringer shape. As shown in Figure 2.2 hat section stringers are the most efficient in carrying axial compression loads. The guidelines for selecting a stringer configuration are identical to those given in Section 2.3.

3.4 PRELIMINARY DESIGN

The design variables calculated in preliminary design are the skin thickness and the stiffener spacing. The design driver is the skin initial buckling load $N_{x,cr}$ and the limiting criteria are the minimum skin thickness

and a reasonable stiffener spacing. Guidelines for the selection of a set of skin thicknesses and, in the case of composites, ply layups for compression panels are identical to those given in Section 2.4.

<u>Calculation of Skin Buckling Strain/Load</u> - The buckling stress for curved metal sheet panels can be calculated from:

$$F_{CR} = \frac{K_c \pi^2 E}{12(1-\nu^2)} \left(\frac{t_w}{b_s}\right)^2 \qquad (10)$$

where,

F_{CR}	buckling stress, psi
t_w	thickness of the skin, in
b_w	effective width of skin panel, in.
E,	modulus and Poisson's ratio for the sheet metal
K_c	buckling coefficient determined from Figure 3.2 (Reference 8)

The theoretical value of K_c is obtained from the buckling equations for thin cylindrical shells and is a function of the nondimensional curvature Z of the panel expressed as

$$Z = \frac{b_s^2 (1-\nu^2)^{1/2}}{r t_w} \qquad (11)$$

where r is the radius of the cylindrical panel. Experimental data have shown that K_c is also a function of the r/t ratio for the panel. The design curves of Figure 3.2, obtained from test data, show this dependence of K_c on r/t.

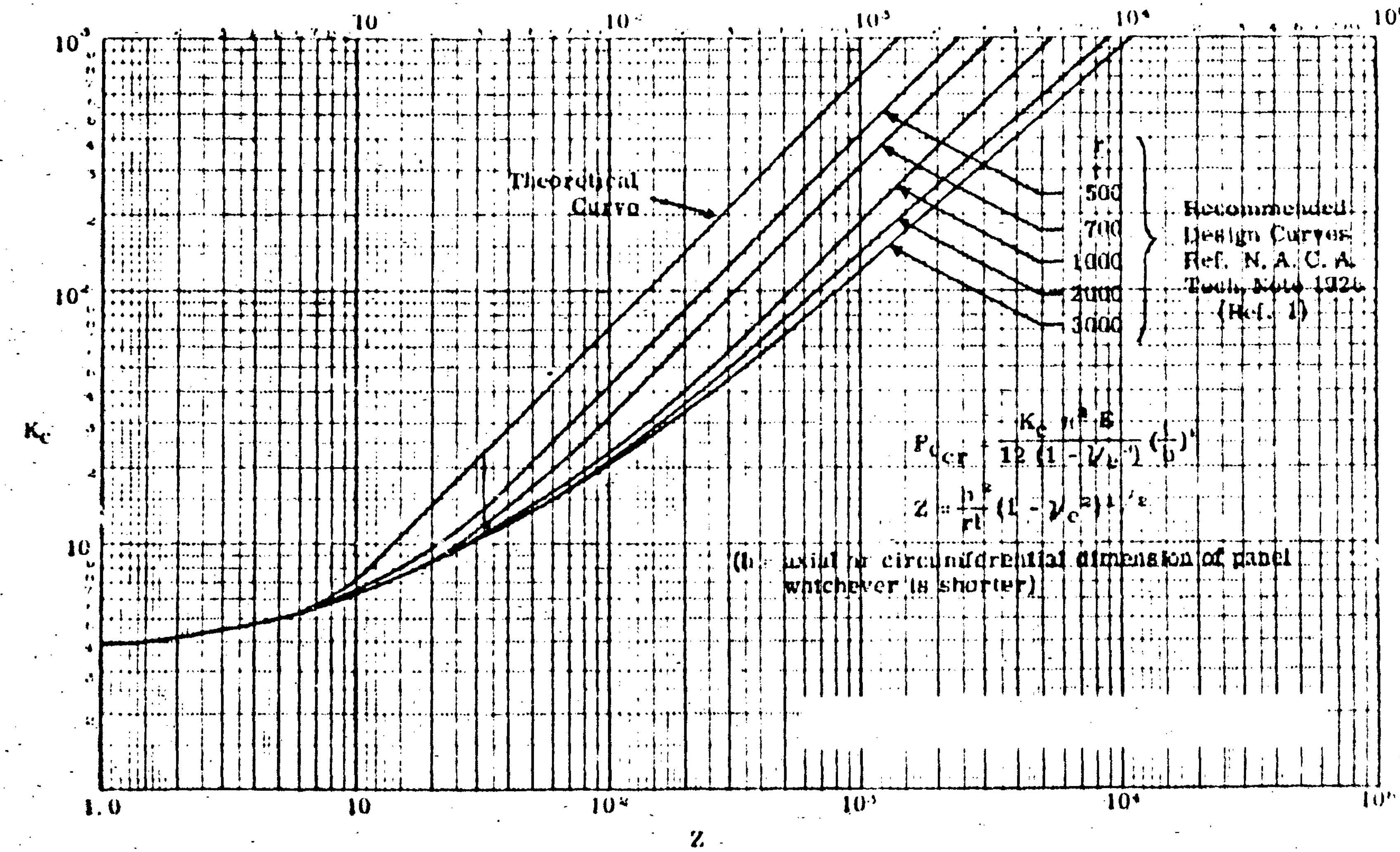


Figure 3.2. Axial Compressive Buckling Coefficients for Long Curved Plates (Reference 34)

Compression buckling strains for curved composite panels can be accurately determined through the use of computer code SS8 (Reference 7), for example. However, for an approximate calculation of the skin buckling strain in cases where the stiffener spacing is realistic, the simplified equation given below has been programmed in CRIP.

$$\varepsilon^{w}_{cr} = \left(\frac{m\pi}{L}\right)^2 \frac{1}{E_{xw}t_w}\left[D_{11} + 2\,(D_{12} + 2D_{66})\left(\frac{nL}{mb_w}\right)^2 + D_{22}\left(\frac{nL}{mb_w}\right)^4\right]$$
$$+ \frac{E_{yw}}{\left(\frac{m\pi}{L}\right)^2 R^2\left[E_{xw} - \left(2\nu_{xyw}E_{yw} - \frac{E_{xw}E_{yw}}{G_{xyw}}\right)\left(\frac{nL}{mb_w}\right)^2 + E_{yw}\left(\frac{nL}{mb_w}\right)^4\right]} \qquad (12)$$

where D_{ij} are the terms of the bending stiffness matrix of the composite skin, E_{xw}, E_{yw}, G_{xyw}, ν_{xyw} and t_w are the web elastic constants and thickness, respectively, L is the panel length, b_w is the effective width of the skin, R is the radius of curvature of the panel and n and m are the integer coefficients representing number of half buckle waves in the width and length direction, respectively. The lowest value of strain for various values of n and m represents the buckling strain of the specimen.

The panel length L corresponds to the frame spacing h_r. The panel effective width b_w equals the stringer spacing h_s for preliminary design. In detail design, however, b_w equals the distance between stringer fastener lines for metal panels and the distance between adjacent stringer flange centerlines as shown in Figure 3.3. For both metal and composite panels the boundary conditions are assumed to be simply supported at the stringers and the frames.

<u>Selection of Skin Thickness and Stringer Spacing</u> - In order to select the skin thickness t_w and the stringer spacing h_s, assuming for the time being that the frame spacing h_r is fixed, plots of calculated buckling loads

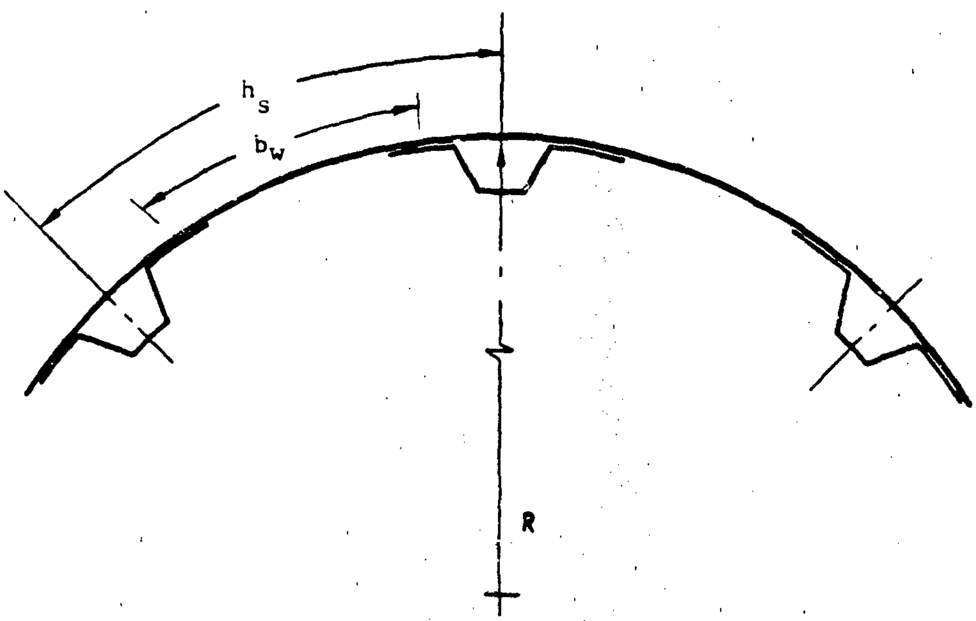


Figure 3.3. Skin Width h_s for Composite Panel Initial Buckling Strain Calculations.

versus the stringer spacing can be generated in a fashion similar to Figure 2.7. From these plots a "best" combination of skin thickness and stringer spacing can be selected.

3.5 DETAIL DESIGN

Detail design of compression panels involves sizing of the stringers and computing margins for the various possible failure modes. The procedure for metal panels is direct in that the required stiffener dimensions are computed for the given load conditions and a standard extrusion selected to match the calculated stringer cross-sectional area. For composite panels the procedure is iterative in that initial sizes are assumed for the stringers, the margins computed, and if any of the margins are negative or too high, the stiffeners are resized and new margins computed. This iteration is continued till all margins are positive and reasonable in magnitude so any weight penalties are minimized.

Initial Dimensions for Composite Stringers - The guidelines to be used are identical to those given for shear panels in Section 2.5.

Stiffener Sizing and Margin Computation - Sizing of the stringers is accomplished using the semiempirical equations given in Reference 1. For composite panels these equations are coded in program CRIP. The use of program CRIP and the sizing procedure for metal panels is outlined in the following paragraphs.

Compression Panel Failure Modes - The various failure modes that have to be checked for in designing compression panels are:

(a) Euler buckling of the stiffened panel,

(b) Stiffener crippling,

(c) Stiffener/web separation in composite panels, and

(d) Permanent set in metal skins.

The load carrying capacity of the panel is then determined as the lowest of the loads calculated for the above failure modes. For composite panels loads corresponding to the stiffener/web separation mode of failure have to be separately calculated to supplement the results of CRIP.

Composite Panels - Program CRIP is applicable to flat as well as curved composite panels. The panel sizing commences by first entering the material properties and the selected skin and stringer dimensions into CRIP. Failure strains corresponding to stringer crippling and Euler buckling are computed next.

The Euler buckling strain is given by:

$$\varepsilon_{CR}^{E} = \frac{C\,\pi^2 EI}{EA\;L^2} \qquad (13)$$

where, EI is the equivalent bending stiffness of the panel, EA is the equivalent axial stiffness, L is the panel length, and C is the end fixity coefficient. The fixity coefficient $C = 3$.

In the event that the Euler buckling margins are not adequate, the I/A or y the distance from the neutral axis to the mid-plane of the skin has to be increased. This can be accomplished by adding more plies to the stringer cap or by increasing the stringer height.

If the margins on stringer crippling are not as desired then the approach to resizing the stringer is defined by examining the equations for stringer crippling strain calculation. The stringer crippling strain is calculated by first modelling the stringer as an interconnected assembly of plate

elements and then calculating the individual plate element crippling strains. These plate elements are sequentially numbered and displayed on the screen during CRIP execution. The stringer crippling strain is the most critical of these plate element crippling strains as described in Reference 1. Thus, to obtain guidelines for stringer resizing, the plate element crippling strain equations have to be examined. These are as follows:

$$\frac{\epsilon_{cs}}{\epsilon_{cr}} = \alpha \left(\frac{\epsilon_{cu}}{\epsilon_{cr}} \right)^{\beta} \tag{14}$$

where,

- ϵ_{cs} = crippling strain of the stiffener plate element
- ϵ_{cr} = initial buckling strain of the stiffener plate element
- ϵ_{cu} = compression ultimate strain for the plate element laminate
- α, β = material dependent coefficients obtained from test data.

The crippling strain for stiffener plate elements connected on both sides is given by:

$$\epsilon_{cs} = 0.56867 \epsilon_{cr} \left(\frac{\epsilon_{cu}}{\epsilon_{cr}} \right)^{0.47567} \tag{15}$$

and ϵ_{cr}, the buckling strain for the stiffener plate element is given by:

$$\epsilon_{cr} = \frac{2\pi^2}{b^2 t\, E_x} \left(\sqrt{D_{11} D_{22}} + D_{12} + 2D_{66} \right) \tag{16}$$

In Equation 15

- b = stiffener plate element width
- t = plate element thickness
- E_x = compression modulus of the plate laminate along the longitudinal direction

D_{ij} = terms from the laminate bending stiffeness matrix, $(i, j = 1, 2, 6)$

Equation 16 applies to plate elements for which the length to width ratio (L/b, where L = stiffener length) is at least 4.

The crippling strain for plate elements that are connected on one side only is calculated using the following equation:

$$\varepsilon_{cc} = 0.4498\varepsilon_{cr}\left(\frac{\varepsilon_{cu}}{\varepsilon_{cr}}\right)^{0.72715} \tag{17}$$

where,

$$\varepsilon_{cr} = \frac{12\,D_{66}}{b^2 t\,E_x} + \frac{4\pi^2 D_{11}}{L^2 t\,E_x} \tag{18}$$

L = length of the stiffener

with the other nomenclature remaining the same as for Equations 15 and 16.

Based on Equations 14 through 18 stringer resizing requires increasing the critical plate element buckling strain. This can be accomplished by:

(a) Increasing the element thickness by adding plies. ±45-degree plies are preferred.

(b) Decreasing the width b of the element.

The new stiffener size, thus, selected is entered in the new run of CRIP and the margins recalculated. The procedure is repeated till the desired margins are achieved.

Stiffener/Web Separation

Failure of composite stiffened panels due to stiffener/web separation is a common mode of failure in the postbuckling range. The stiffener/skin separation strain for cocured AS/3501-6 panels is calculated from:

$$\varepsilon_{SS} = 0.4498\ \varepsilon_{cr} \left(\frac{\varepsilon_{cu}}{\varepsilon_{cr}}\right)^{.72715} \tag{19}$$

where,

ϵ_{ss} = stiffener/web separation failure strain

ϵ_{cu} = compression ultimate strain for the plate element cocured with the skin

ϵ_{cr} = skin buckling strain.

This failure mode check is not coded in CRIP.

Metal Panels

The Euler buckling strain for metal panels is calculated using Equation 13.

Stiffener Crippling Strain/Stress Calculation - The crippling strength of metal stiffeners is calculated using the well established Gerard

method documented in Reference 8. The empirical Gerard equation for calculating the crippling stress for 2 corner sections, such as the Z, J, and channel sections, is:

$$\frac{F_{cs}}{F_{cy}} = 3.2\left[\left(\frac{t^2}{A}\right)\left(\frac{E}{F_{cy}}\right)^{1/3}\right]^{0.75} \tag{20}$$

where,

F_{cs} = crippling stress for the section, psi

F_{cy} = compressive yield stress of the material, psi

t = element thickness, in

A = section area, in^2

A design curve based on Equation 20 is shown in Figure 3.4 taken from Reference 8. Additional crippling equations that apply to sections other than 2 corner sections are given in Reference 8.

Failure Load Calculation - The failure load is calculated as the lowest of the loads corresponding to Euler buckling and stiffener crippling.

For Euler buckling, the failure load is given by:

$$P_E = \epsilon_{cr}^{E} \left(E_{xs}A_s + E_{xw}b_w t_w\right) \tag{21}$$

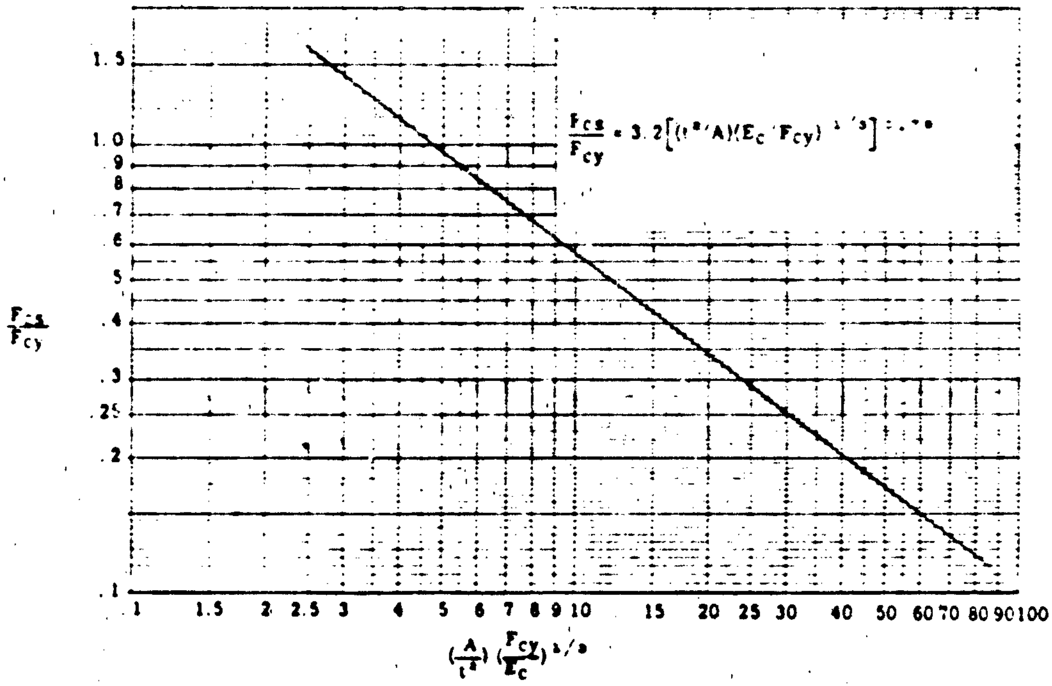


Figure 3.4. Crippling Stress F_{cs} for Two Corner Sections Z, J and Channel Sections (Reference 34).

where,

ϵ_{cr}^{E} = Euler buckling strain calculated from Equation 13

A_s = Stringer cross-sectional area

h_s = Stringer spacing

t_w = Skin Thickness

The running load is given by:

$$N_x^E = \frac{P_E}{h_s} \tag{22}$$

The margin is computed as:

$$(M.S.)_{Euler} = \frac{N_x{}^E}{N_x{}^{DUL}} - 1 \tag{23}$$

where, $N_x{}^{DUL}$ is the design ultimate load.

The failure load due to stringer crippling is given by:

$$P_{cc} = F_{cs}A_s + F_{cs}\,wt \tag{24}$$

where,

F_{cs} = Stiffener crippling stress

$w = 1.9\, t_w \sqrt{\frac{E}{F_{cs}}}$ is the effective skin width

A_s and t are as defined for Equation (21)

The margin is computed as:

$$(M.S.)^{cc} = \frac{N_x^{cc}}{N_x^{DUL}} - 1 \qquad (25)$$

3.6 EXAMPLES

The composite and metal compression panel design procedure outlined above is demonstrated by way of the following examples.

Example 1. Curved Composite Panel

A postbuckled composite compression panel with a 45 in. radius and 20 in. frame spacing (h_r) is to be designed to carry a design ultimate compression load of 900 lbs/in. Skins are not permitted to buckle below 25 percent of the design limit load.

Design Procedure

(a) Design Criteria:

The material selected is AS/3501-6 unidirectional graphite/epoxy with the following properties.

$E_1 = 17.6 \times 10^6$ psi $E_2 = 1.9 \times 10^6$ psi $G_{12} = .85 \times 10^6$ psi

$\nu_{12} = 0.3$

The compression ultimate strains are:

$\epsilon_{cu} = .012$ for laminates with at least 40% 0-degree plies

$= .015$ otherwise

Loads:

Design Ultimate Load, N_x (DUL) = 900 lbs/in

Design Limit Load (DLL) = 600 lbs/in

Initial Buckling Load (IBL) = 150 lbs/in

(b) Configuration Selection:

Panel Radius, R = 45 in.

Frame Spacing, h_r = 20 in.

} Given

Skins designed for buckling only

Select [±45/∓45] layup

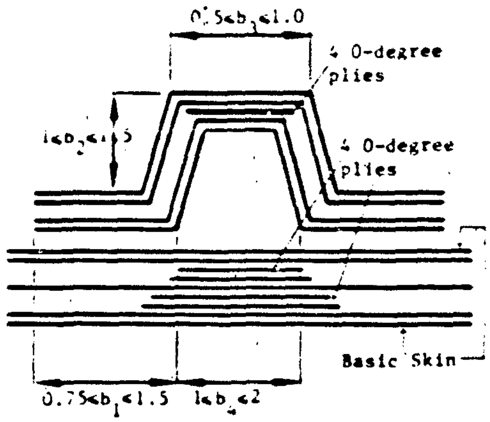
Select hat section stringers for efficiency.

(c) Preliminary Design:

To meet initial buckling load requirement of 150 lbs/in with the given layup, stringer spacing h_r = 8.2 in.

(d) **Detail Design:**

(1) Select initial stringer dimensions and ply distribution using the guideline of Figure 2.10 and previous experience.



Initial stringer dimensions:

ELEMENT NO.	WIDTH, IN.	LAYUP
1	$b_1 = 1.2$	$[\pm 45]_{2s}$
2	$b_2 = 2.0$	$[\pm 45]_s$
3	$b_3 = 0.6$	$[\pm 45/0_2]_s$
4	$b_4 = 1.2$	$[\pm 45/0_2]_s$

(2) Run program CRIP for margin computations.

CRIP output for initial stringer dimensions: Skin buckling strain = .000440

PROPERTY	OUTPUT
Euler Buckling Strain	.061356
Euler Buckling Load, lbs	97,720
EA, lbs	$.131 \times 10^7$
EI, lbs-in^2	$.898 \times 10^6$
y, in	.574
Crippling Strain*	.002267
Crippling Load, P_{co}, lbs	3092
Crippling Load $N_x = P_{co}/8.2$, lbs/in	377
Margin, %	-58
Load in Stiffener at failure, %	96

*ELEMENT 2 IS CRITICAL

The negative margin on stringer crippling requires that element 2 be shortened and load redistributed from this element to the caps and skins under the cap by adding 0-degree plies.

(3) The stringer is resized as follows for the 2nd iteration:

ELEMENT NO.	WIDTH, IN.	LAYUP
1	$b_1 = 1.0$	$[\pm 45]_{2s}$
2	$b_2 = 1.3$	$[\pm 45]_s$
3	$b_3 = 0.6$	$[\pm 45/0_8]_s$
4	$b_4 = 1.2$	$[\pm 45/0_3]_s$

(4) Run program CRIP for margin computations. CRIP output is as follows: Skin Buckling Strain = .000440.

PROPERTY	OUTPUT
Euler Buckling Strain	.03727
Euler Buckling Load, lbs	89,750
EA, lbs	$.209 \times 10^7$
EI, lbs-in^2	$.791 \times 10^6$
y, in	.61
Crippling Strain*	.0035
Crippling Load, P_{cc}, lbs	7401
Crippling Load $N_x = P_{cc}/8.2$, lbs/in	902.6
Margin, %	0
Load in Stiffener at failure, %	98

*ELEMENT 2 IS CRITICAL IN CRIPPLING

The predicted failure mode is stringer crippling.

Example 2. Curved Metal Compression Panel

A postbuckled curved aluminum panel of 45 in. radius, 10 in. stringer spacing (h_s) and 20 in. frame spacing (h_r) is to be designed to carry axial compression loading. The design ultimate load is 900 lbs/in. The skins are not permitted to buckle below 33 percent of the design limit load.

Design Procedure:

(a) Design Criteria:

Material: 7075-T6

Properties: $E_c = 10.6x10^6$ psi, $= 0.3$

Yield Strain: .0061

Loads:

Design Ultimate Load, N_x (DUL) = 900 lbs/in

Design Limit Load (DLL) = 600 lbs/in

Initial Buckling Load (IBL) = 200 lbs/in

(b) Configuration Selection:

Panel radius, R = 45 in

Stringer spacing, h_s = 10 in.

Frame spacing, h_r = 20 in.

Given

Select skin thickness, t_w = 0.05 in.

Select Z-section stringers

(c) Detail Design:

Initial buckling stress:

$$F_{cr} = \frac{K_c \pi^2 E}{12(1-\nu^2)} \left(\frac{t_w}{b_s}\right)^2$$

where $K_c = 13$ (from Figure 3.2)

$$F_{cr} = 3143 \text{ psi}$$

Stringer Sizing:

The design buckling load ($N_{x,cr}$), however, = 200 lb/in. Hence, the stiffener area (A_s) required is obtained as follows:

$$A_s = \frac{N_{x,cr}\ b_s}{F_{cr}} - b_s\ t_w$$

$$A_s = .1363 \text{ inch}^2$$

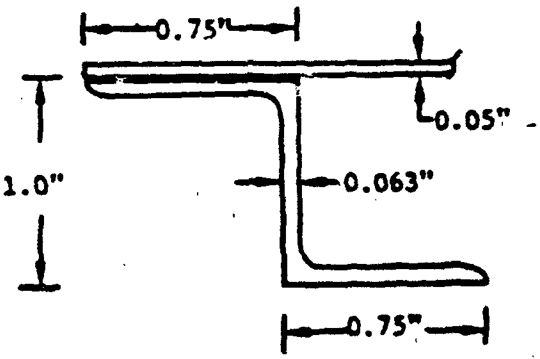
Assume the stiffener configuration to be AND 10138-1C04 which is shown in Figure 3.5. Thus, the predicted panel buckling load $N_{xy,cr} = 206$ lb/in which results in a 3 percent margin of safety.

Figure 3.4 was used to calculate the crippling stress F_{cs} for the stiffener:

$$F_{cs} = 48.9 \text{ ksi}$$

The effective web width at the time of stiffener crippling, w, was calculated from Equation 24 as:

$$w = 1.2 \text{ inch}$$



stiffener area A_s = 0.155 sq. inch

stiffener M.O.I. I_{xx_s} = 0.0236 inch^4

Figure 3.5 Z-Section 7075-T6 Aluminum Stringer. AND 10138-1004 Configuration.

The total load at panel failure P_{ult} is calculated using Equation 24 as:

$$P_{ult} = F_{cs} (A_s + w\ t_w)$$
$$= 48900\ (.155 + 1.2 \times .05)$$
$$P_{ult} = 10500\ \text{lb.}$$

Hence, the ultimate failure load per unit width ($N_{x_{ult}}$) is

$$N_{x_{ult}} = \frac{P_{ult}}{b_s} = 1050\ \text{lb/in}$$

Thus, the panel failure load allows approximately a 17 percent margin of safety.

For overall panel instability the Euler buckling stress was calculated using Equation 13 in the following form:

$$\sigma_{Euler} = \frac{\pi^2\ EI_e}{L_e^2\ A_t}$$

where, L_e is the effective length of the panel, A_t is the total area of the panel and I_e is the panel moment of inertia about the neutral axis. Since the frame spacing for design purposes was assumed to be 20 inches, the effective length "L" for Euler buckling is 10 inches (C = 4 in., Equation 13) assuming fully fixed ends. Thus, the calculated Euler buckling stress for the panel was:

$$\sigma_{Euler} = 90.63\ \text{ksi}$$

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